No: 9/90 | Ref: EW/C1166 | Category: 1a

Aircraft Type and Registration:	Lockheed L-1011-385-1 Tristar, N31019
No & Type of Engines:	3 Rolls-Royce RB 211- 22B turbofan engines
Year of Manufacture:	1974
Date and Time (UTC):	15 June 1990 at 0948 hrs
Location:	London Heathrow Airport
Type of Flight:	Public Transport
Persons on Board:	Crew - 15 Passengers - 222
Injuries:	Crew - 1 (minor) Passengers - 3 (minor)
Nature of Damage:	Damage to bleed air ducting, insulation and cabin furnishings
Commander's Licence:	Airline Transport Rating (FAA)
Commander's Age:	51 years
Commander's Total Flying Experience:	18,000 hours (of which 400 were on type)
Information Source:	AAIB Field Investigation

History of the Flight

The aircraft, which was bound for New York, started up and taxied out without incident to the threshold of runway 27R. Following Air Traffic Control (ATC) clearance to take-off, engine power was applied and the aircraft began to accelerate. A few seconds later, at about 20 kts, the flight deck crew head a muffled 'thump' and felt a pressure surge within the aircraft. Although there were no immediate cockpit indications of a problem, the commander rejected the take-off and brought the aircraft to a standstill. A few seconds later, the 'Overheat in area J' warning light illuminated on the engineer's panel and the commander ordered the engine bleed air system in that area be isolated.

As the commander was applying power to leave the runway, at the request of ATC, the cabin staff informed him that there had been a loud bang and a flash and that the rear cabin was full of smoke. Fearing that there might be a fire on board, the aircraft was stopped on the runway and an emergency evacuation was initiated. With the exception of the two rearmost slides on the left side, which were deliberately not used, all emergency escape slides were successfully deployed. Several passengers sustained minor injuries as a result of the evacuation.

Both the Flight Data Recorder, a Sundstrand DFDR with ARINC 563 recording format, and Cockpit Voice Recorder, a Fairchild A100 model, were removed from the aircraft and taken to the AAIB replay facility at Farnborough. A satisfactory replay was obtained from both. The DFDR showed the recorded engine pressure ratio (EPR) had increased to a maximum of 1.34 before reducing, as the take-off was abandoned. It then showed another smaller increase in EPR some 40 seconds later, before the engines were shut down and the recording ceased.

Aircraft examination

Initial concern that an explosive device had been on board proved groundless and subsequent inspection quickly revealed that the No 2 engine bleed air duct thermal expansion compensator had suffered complete failure of a circumferential welded joint. This unit was located on the left side of the rear fuselage, beneath the cabin floor, below seat row Nos 33 and 34. The immediate result of this failure was to discharge high temperature, high pressure bleed air (maximum 50 psi and 485 deg F) from the 8 inch diameter duct into the area between the rear cargo hold liner and the fuselage skin (area J), which then found escape paths into the rear cargo holds and up into the passenger cabin via the air conditioning vents. No mechanical damage had been caused to the aircraft's primary structure, but damage was present on the bleed air ducting remote from the location of the failure, to some related secondary support structure and cargo hold liner panels. Due to the relatively short time scale involved, there had been no apparent overheating of the local airframe, hydraulic pipes or wiring looms, although several insulation blankets had ruptured, releasing large quantities of fibrous material into the cabin. In the passenger cabin, two sidewall panels adjacent to seat row No 34 were dislodged, the rear toilet door had been forced open and most of the left side overhead luggage locker doors in the rear cabin had sprung open. Although passengers and cabin crew had reported seeing a flash and temporary reduced visibility due to 'smoke', no evidence was found of fire, sooting or smoke damage on the aircraft.

Compensator history

The compensator that failed, Part No 1506374 - 102, was manufactured from thin titanium sheet by Stainless Steel Products (SSP) and this standard (-102) of the compensator was the subject of a manufacturer's Service Bulletin (SB), No 6374-36-01, which was issued in November 1977. At that time several -101 and -102 standard compensators had suffered complete failures of the same particular welded circumferential joint, on sub assembly 1506374-93. The SB addressed this problem by increasing the size of the weld and section 1B, of the SB, states that these failures were 'not of a fatigue type'.

This particular compensator was fitted to the aircraft on 5 March 1987 since when it had accumulated 12,085 hours and 3,325 flights. These items are removed at a major check, not to exceed 13,500 hours, when they are sent to the workshop for a crack inspection of the welds. At check C intervals, not to exceed 2,500 hours, the ducts are inspected visually on the aircraft. However, as a result of a similar duct failure which occurred at Los Angeles on another Tristar operated by the same airline, on 18 February 1990, all such compensators were removed for a one-off special dye penetrant crack check.

Compensator Examination

The two sections of the failed compensator were taken to the Materials Dept of the Royal Aerospace Establishment at Farnborough where a detailed examination of the fracture surfaces was carried out. The compensator body is shown in Fig 1 with the position of the circumferential fracture arrowed.The barrel assembly, Fig 2, of which the flange was an integral part, extended into the body and acted as the inner part of a sliding seal and, before failing, had been attached to the body by the circumferential TIG (Tungsten Inert Gas) weld at the position arrowed. A length of this weld and the associated fracture are shown in detail in Fig 3, while Fig 4 illustrates the position on the body where radial tearing was evident and where it is considered that final fracture took place. This corresponds to a step in the fracture path arrowed in Fig 3. There were two other circumferential resistance welds in the vicinity of the failure, as can be seen in Fig 3, and it was evident that the local construction was quite complex.

The fracture, when viewed at low magnification, was very facetted along the whole of its length and appeared to contain a lot of cleavage-type rupture, with little evidence of ductility. Apart from the short length over which final failure occurred there was no distortion of the fracture around the inner sliding member and no other evidence that overstressing had been involved in the separation.

Metallography

A section, parallel to the longitudinal axis of the duct, was cut through the fracture and the adjacent welds and is shown in Fig 5, to illustrate the mode of construction in that region. The original thickness of the sheet that fractured at the weld, arrowed A, was obtained from this section and found to be 0.75 mm, although clearly it had been thickened by TIG welding at the position of failure.

Comparison of the section shown in Fig 5 with a detail of the local construction taken from Welding Procedure Drawing No 250 (part of the SB) for the compensator, Fig 6, indicated that the configuration of the particular joint of the compensator that had failed was not in accordance with the design shown in the SB or assembly drawing. These drawings showed that the failed part of the assembly should have been bent from a slope of 60° to run parallel to the longitudinal axis adjacent to the weld. However, in the section, it is evident that it actually met the attachment at 60°.

The heat affected zone of the TIG weld extended for a considerable distance either side of the actual joint, as indicated by the presence of coarse grained microstructure in that region. This accounted for the facetted appearance of the fracture and indicated that excessive heating had occurred during welding. At high magnification it was evident that the section immediately adjacent to the weld contained several internal cracks, which are shown in detail in Fig 7. The position and general appearance of these cracks was not consistent with fatigue but suggested instead a condition of stress cracking that can occur in the region of a weld during cooling.

Separation of the inner sliding sleeve from the body of the compensator occurred due to the growth of fatigue cracks around the whole of the circumferential attachment weld. Fractographic evidence showed that crack growth had taken place from the inside of the welded joint and suggested that some of the cracking had initiated at the inner surface. However, the sheet titanium contained cracks within the section adjacent to the fracture and these cracks had the appearance of stress cracks of the type produced when a weld cools under constraint. It seems likely that some of the fatigue cracks developed internally from these pre-existing cracks and that the stress cracks were a significant factor in the fatigue failure. Although it is not known precisely what stresses act on the failure region in service, it is assumed that they are generated by the effects of pressurisation, thermal expansion and contraction, and by resistance of the internal seals to sliding under the influence of expansion and contraction of the duct, for which the compensator is designed.

It is likely that, due to the configuration of the failed joint, the cooling contraction stresses concentrated at the weld would have been significantly higher than they would have been if the joint had been made in the manner shown in the welding procedure drawing, the same effect being likely to occur on post SB compensators of the same joint configuration

Currently the FAA are proposing to issue an Airworthiness Directive, requiring the inspection, within 300 hours, of early standard compensators for cracks in this weld, with subsequent inspections every 150 hours, up to a maximum of 1800 hours from the implementation date of the AD. After that time it is proposed that later standard compensators must be fitted. The airline concerned have stated that by 27 June 1990 only compensators that are in compliance with the SB will be installed in the aft duct location on their aircraft.

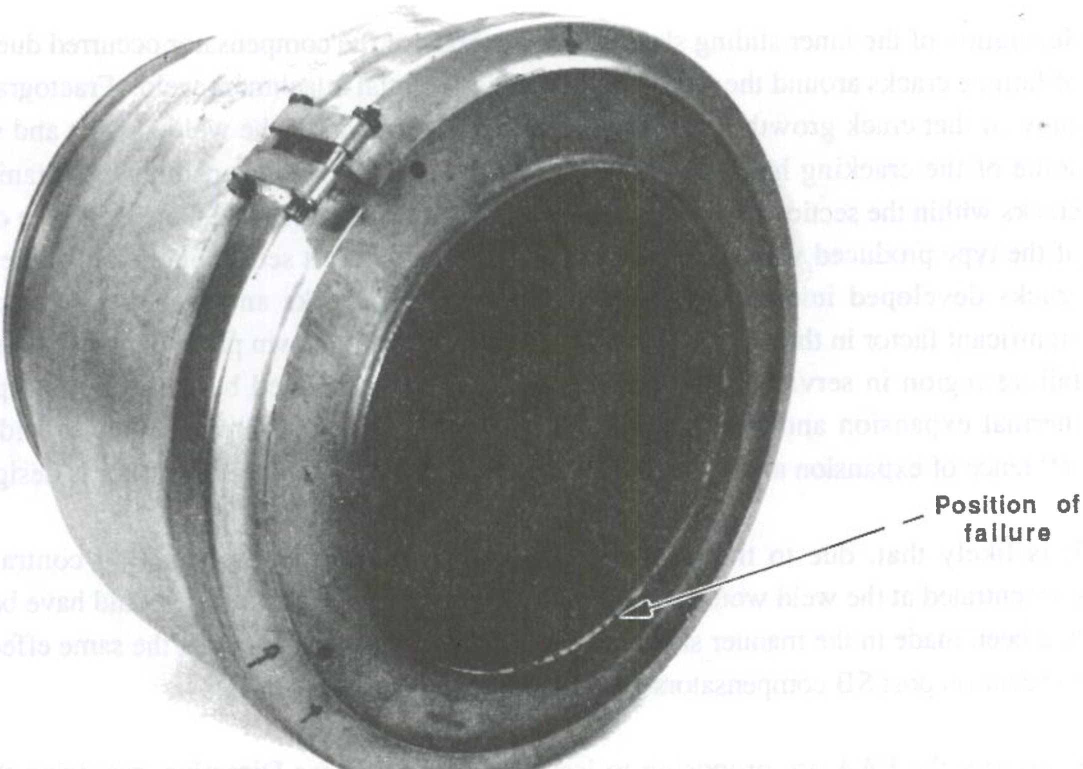


Fig 1 Compensator body

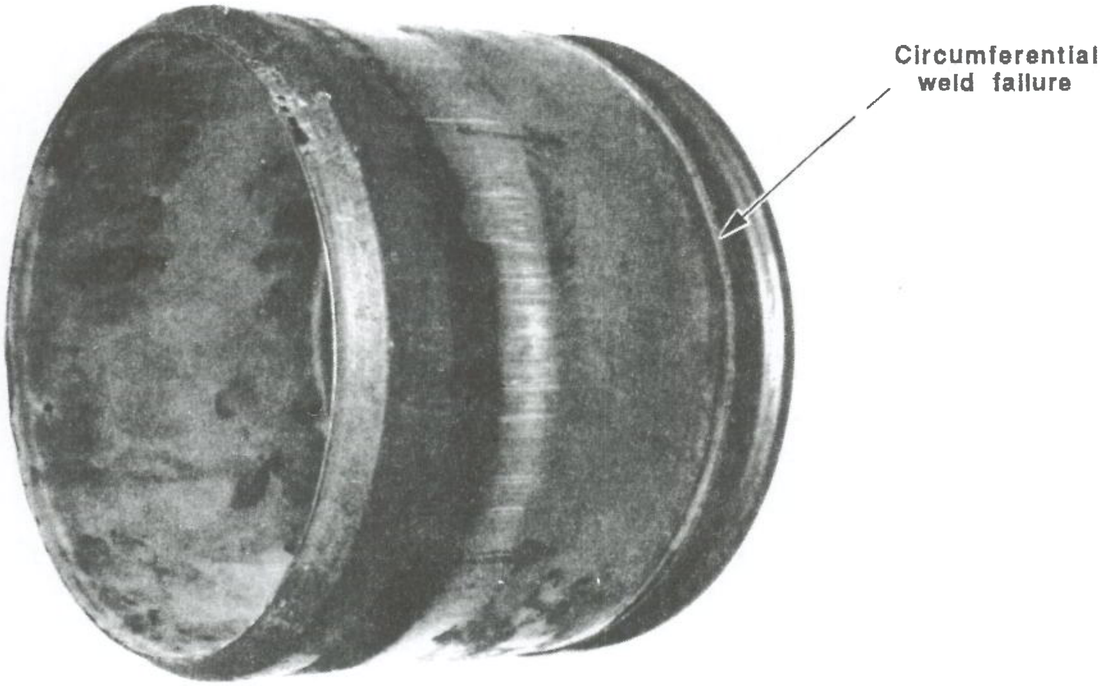


Fig 2 Barrel assembly

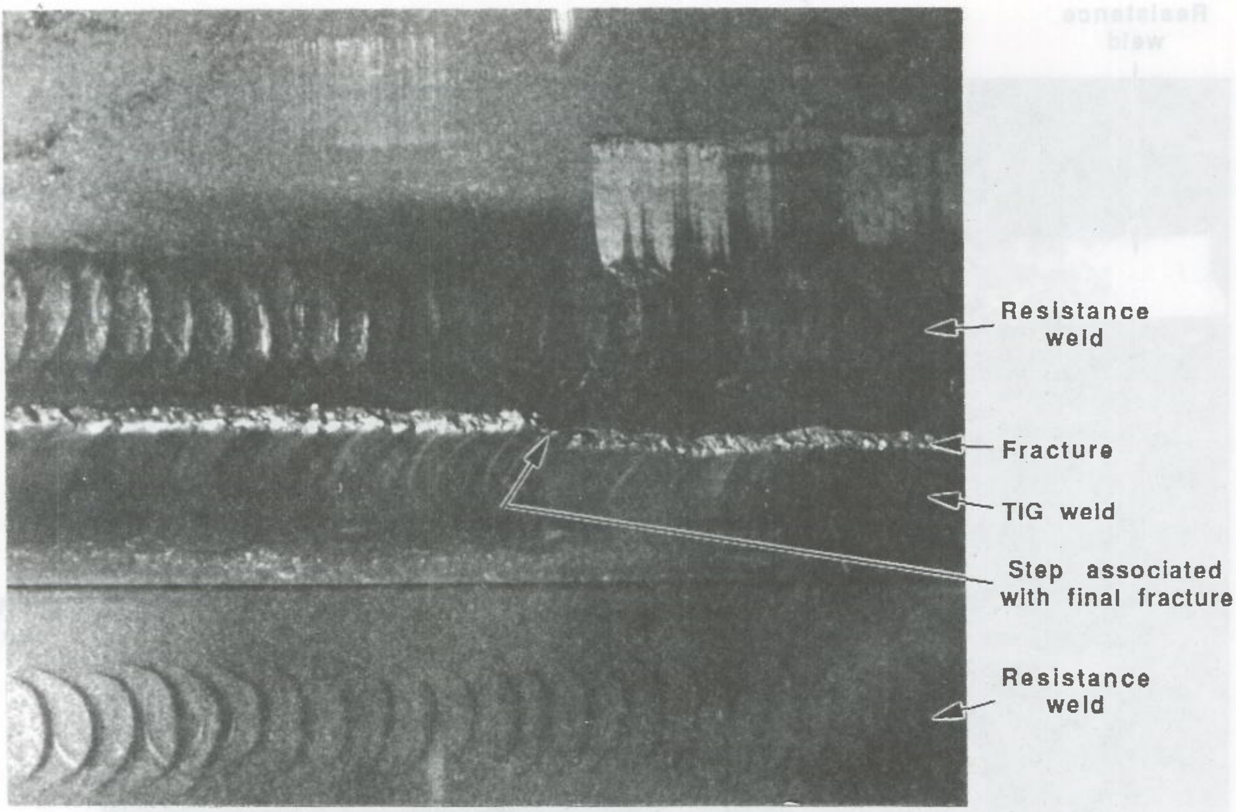


Fig 3 Detail of fracture associated with circumferential weld around inner sleeve

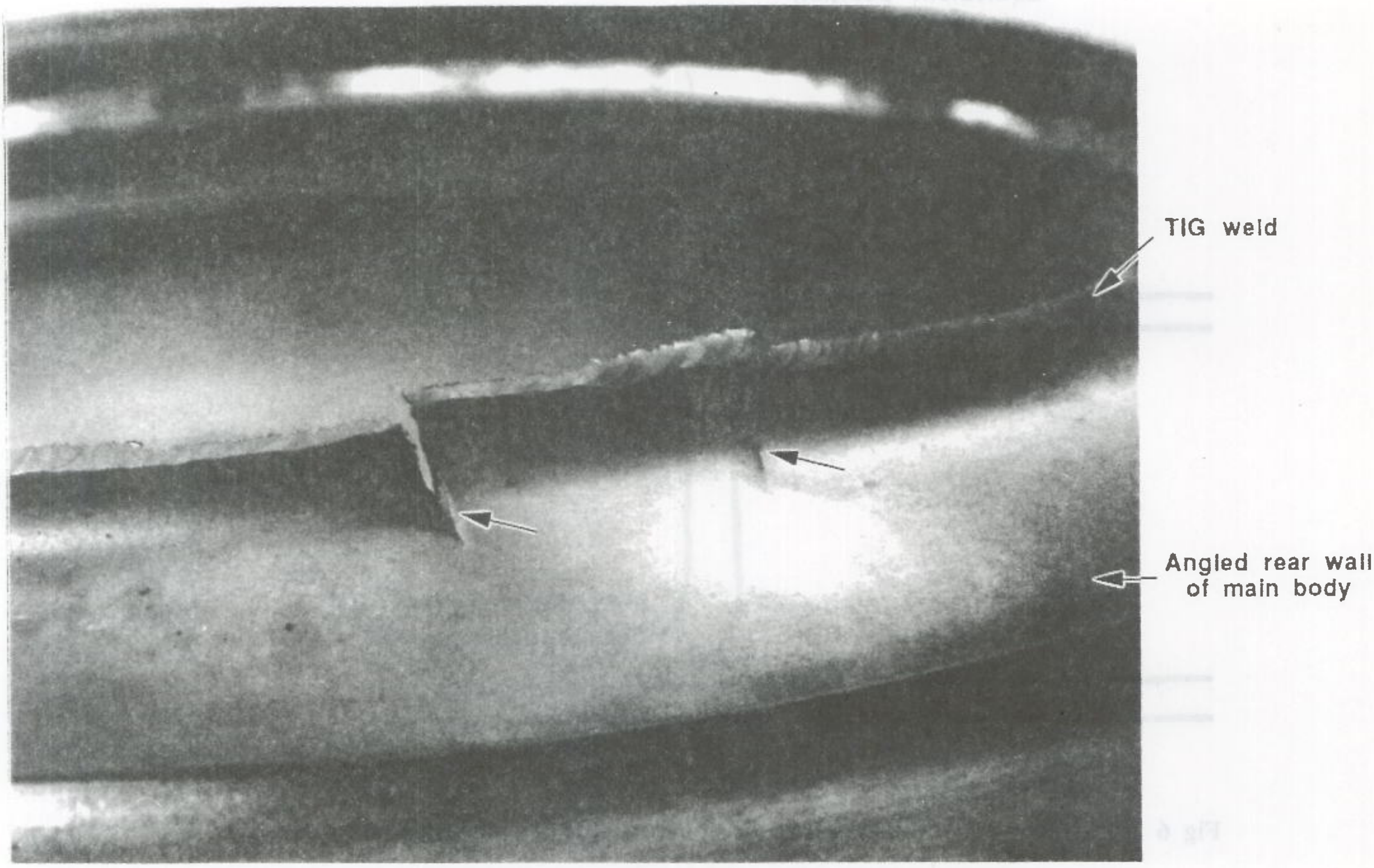


Fig 4 Radial tearing associated with final fracture

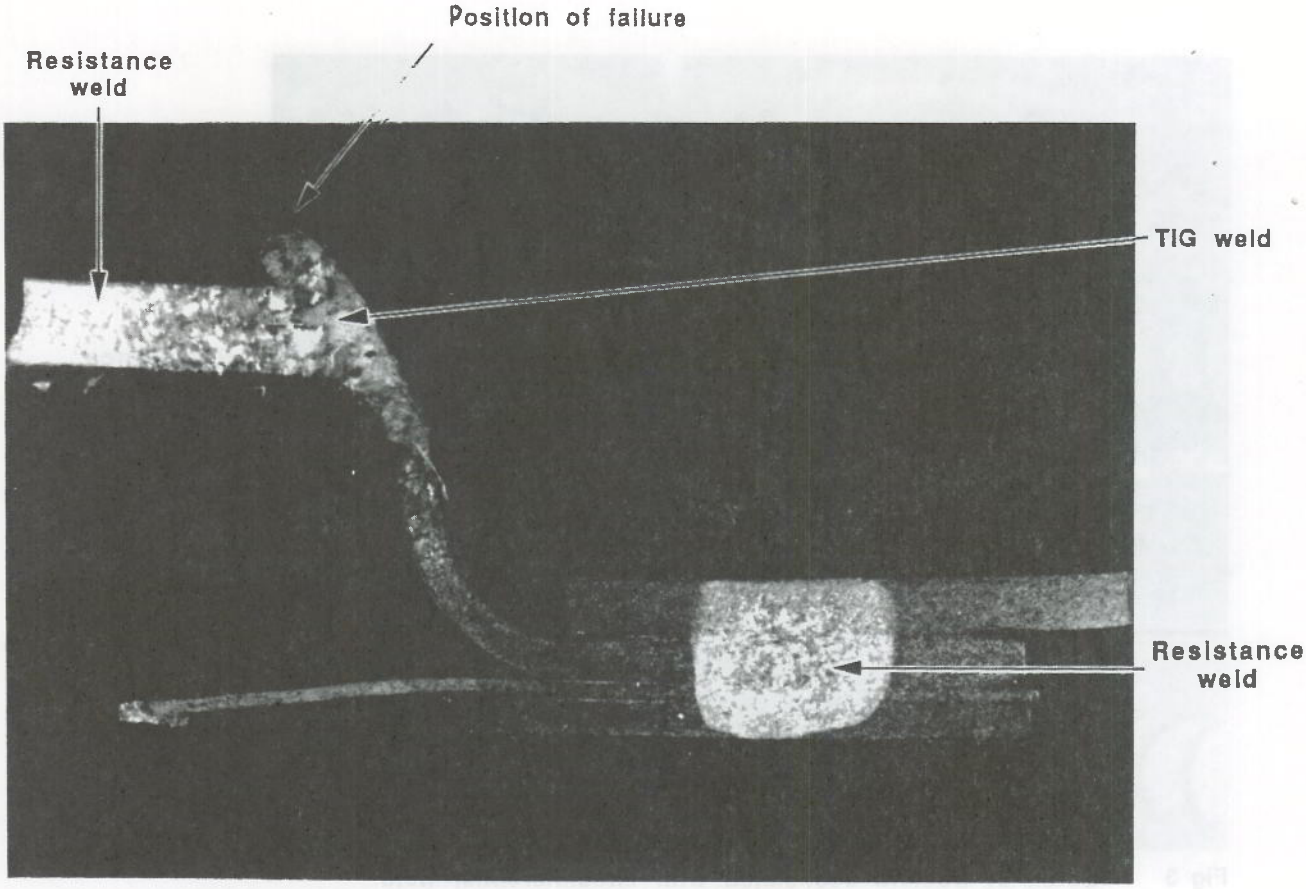


Fig 5 Longitudinal section through structure in region of fracture A X6

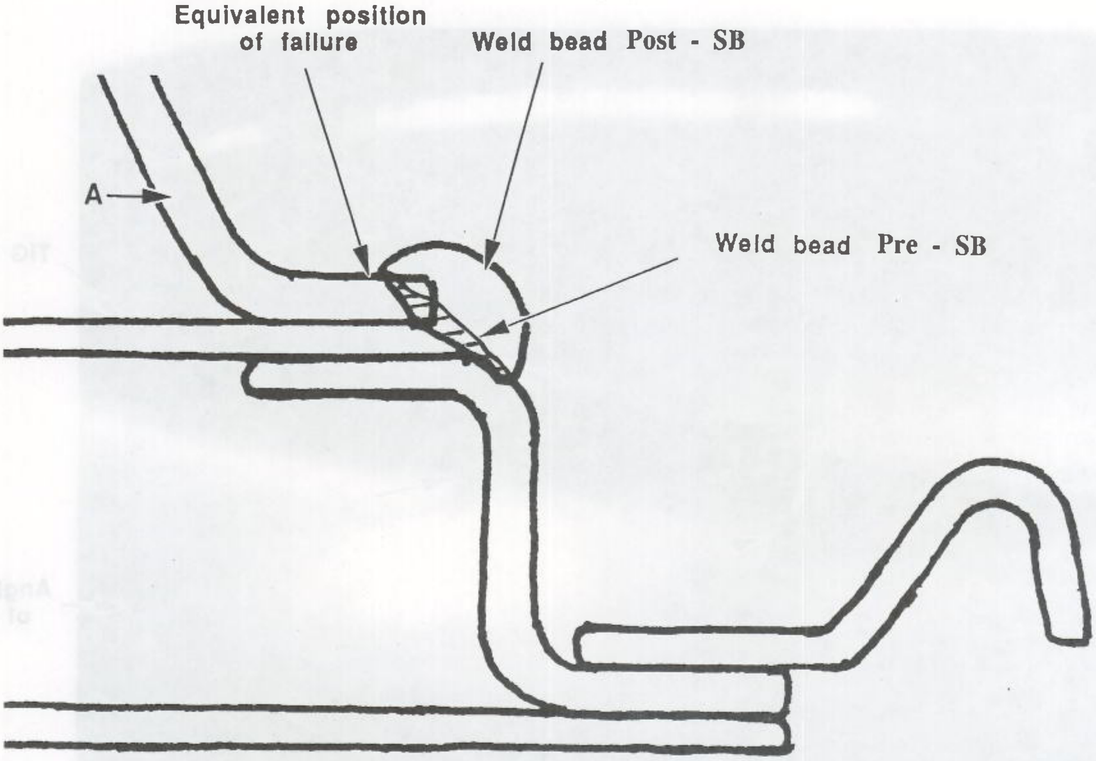
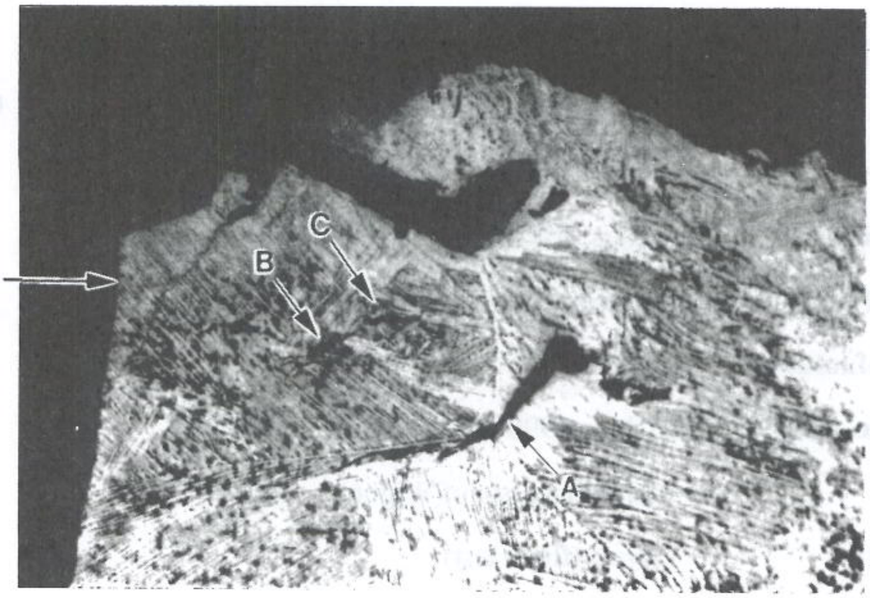


Fig 6 Detail from welding procedure drawing



Examples of internal cracks adjacent to fatigue fracture X100

Detail of crack A X200

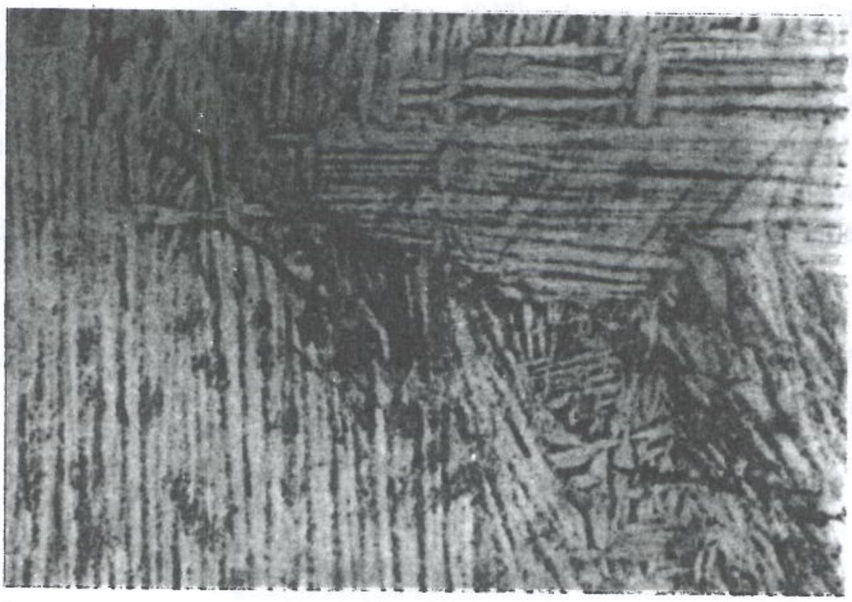


Fig 7 Detail of multiple cracks at B X500